REPORT No. 76

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FOR AERONAUTICS

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BY

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Langley Field, Va.

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FUSELAGE STRESS ANALYSIS.

By Edward P. Warner and Roy G. Miller.

INTRODUCTION.

There is, at the present time, a wide diversity in the methods employed by designers for analyzing the stresses in a fuselage of the built-up type in which the shear is taken by diagonal bracing wires, and there is a similar diversity of opinion as to the best type of analysis manifested in the standards and specifications issued by various governmental agencies.

In a specification for pursuit machines (No. 1003) issued by the Signal Corps just after the United States entered the war, the requirement is that the fuselage be designed to stand a dynamic load factor of 5 and a tail load of 27 pounds per square foot for machines having a maximum horizontal speed of 100 miles per hour, 38 pounds per square foot where the maximum horizontal speed was 120 miles per hour, and so on for higher or lower speeds, the load per unit of area varying as the square of the speed. For machines making 100 miles per hour this corresponded to a tail load of 5.4 pounds per square foot for each unit of dynamic load. This is very nearly equal to the relation between these two loads now laid down in the Air Service's specifications for fuselage sand load tests, where it is prescribed that the tail load shall be 5 pounds per square foot for each unit of dynamic load. A fuselage should, with this type of loading, stand a dynamic load of at least 4, with 20 pounds per square foot on the tail. The general specifications of the Bureau of Construction and Repair, United States Navy, fix the required tail load at 20 pounds per square foot, but combine with this only the weight of the machine without a dynamic factor.

These specifications relate primarily to flying loads. In studying the distribution of load on landing, also, there is some difference of opinion as to the best method of procedure, although the vexed question of tail loading is avoided in this case. Obviously there are many different ways in which a landing may be effected, and the distribution of the landing loads, as well as their magnitude, will vary with the speed of the machine, its attitude at the instant of touching the ground, the position of the controls, the nature of the ground, and a number of other factors.

The method adopted in this report is, in the main, the straightforward one of choosing a standard type of machine, analyzing it by many different methods, and comparing the results. The loadings for all of the seven cases studied are illustrated in figures 1 and 2, and the stresses in the members for all the loadings tried are tabulated at the end of the report.

In the tabulation, the largest stress occurring in each member has been printed in italics. Where the same member may take tension or compression under different loadings, the maximum stress of each sort is printed in italics. In picking out the maxima, Case VII was ignored for reasons which will be apparent later. In some cases it has been possible to make results clearer by analytical discussion, but the graphical analysis has been carried through in every case. The dynamic factor used is 5 in all cases, this being considered to allow for very heavy landings and also for the worst loads experienced in flying. If the sizes of members were being selected, a further "material factor" or true factor of safety of about 1½ should be allowed over the tabulated stresses (i. e., the ultimate strength of each member should be at least 1½ times the largest stress set down against that member in the table).

If a really thorough and accurate analysis of the stresses in an airplane were to be made it would be necessary to treat the whole structure as a unit instead of separating wings and fuselage as is the invariable custom. The drag wires and other members so inter-connect the parts of the machine that no single part can be treated independently with strict accuracy. So many uncertainties arise, however, chiefly due to the presence of redundant members, the stress in which is determined by the rigidity of their end connections and by the initial tension with which they are set up, that it is usually best to consider these redundant inter-connecting members as additional safeguards, and not to take them into consideration in figuring stresses. There are some conditions, however, as will be seen in connection with Case VI, where it would be manifestly absurd to ignore entirely the effect of the external drag wires on the fuselage stresses.

The fuselage adopted is similar to that of the JN4H. The layout of the center lines of the members is identical with that in the JN4H, but the load distribution differs in some particulars from that adopted in the Curtiss design, and the stresses given in the report therefore should not be taken as representing those actually existing in the JN.

LANDING LOADS.

The landing stresses have been examined on four different sets of assumptions, three relating to landings with the tail high, the fourth to those with the tail low (three-point). The lift of the wings has been neglected in all cases. Its inclusion or omission has little effect on any stresses except those in the chassis struts and in the vertical members directly over them. If the weight of the airplane be broken up into loads applied at the panel points in the usual manner, and if the sum of these loads be opposed by a vertical force acting at the axle, which is manifestly the only place that it can act until the tail skid touches the ground, the resultant of the loads at the panel points will pass through the center of gravity, while the vertical through the axle will pass well forward of this point. Therefore, although the two resultant forces are equal in magnitude and opposite in direction they are not directly opposed, and their resultant is a turning moment which, if a landing were actually made under these conditions, would whip the tail of the machine violently downward. Actually, however, the force applied at the axle is not vertical, but is inclined backwards by the frictional resistance to forward motion. This frictional resistance varies widely with the nature of the ground, but it is common to arbitrarily assume, in order to simplify the analysis, that it is just sufficient to incline the line representing the resultant ground reaction until it passes through the center of gravity. If this assumption be made the two resultant forces (ground reactions and resultant of loads at panel points) no longer give a moment, as they both pass through the center of gravity, but they are no longer directly opposed, and there is an unbalanced horizontal force which must be taken care of in order that complete statical equilibrium of the structure may be secured. The securing of statical equilibrium is, of course, an essential condition for the closure of any stress diagram. The horizontal component of the ground reaction has been treated in two different ways (Cases I and III). In the first, it is assumed that this force is not balanced by any external force, and that the machine is therefore decelerating. The deceleration of the several weights applied at the panel points tends to throw them forward, and the resultant of the weight and the inertia force lies in a line parallel to the resultant ground reaction. Since the vertical component of each force remains unchanged, the load at each panel point should be multiplied by the secant of the angle through which its line of action is turned by the inertia force. This method of applying landing loads was first suggested and applied by J. A. Roché.[1] The distribution between the upper and lower panel points of the parts of a load is inversely proportional to the distances from the load to the panel points (just like the supporting reactions for a simple beam). This is rigorously correct for the horizontal components, but not for the vertical ones, since the end fixtures of fuselage struts usually will not transmit tension, and vertical loads applied between the ends of the strut are transmitted to the longeron entirely at the lower end of the strut. It was shown in Mr. Roché's article that the maximum stresses are not changed in any member if the vertical and horizontal loads are both divided between the upper and

[1] Methods Used in Finding Fuselage Stresses, by J. A. Roché: Aerial Age, July 23, 1917, vol. 5, p. 642.

lower panel points in the proportion correct for the horizontal ones, and such an arrangement of loads, making the lines of action of all the forces parallel to each other, greatly simplifies the work. The second method is to balance the horizontal component of ground reaction directly by a thrust load. This is the condition in taxi-ing, and is illustrated in Case III. The assumption that the thrust is large enough to balance completely the horizontal component while the dynamic load is a maximum is rather severe, as the thrust would have to be about equal to the weight of the machine when the dynamic load was 5. The figures obtained on this assumption, however, at least serve to show in which members the stress is increased by a thrust load.

Case II is intended to relate to the same condition as Case I, but in a simplified form making it unnecessary to deal with inclined loads. Instead of taking the reactions at the points of attachment of the chassis strust as passing along the struts, so that their resultant might pass through the axle, the resultant of the panel-point loads is opposed, in Case II, by a vertical force passing through the center of gravity. The forces in the chassis struts are then arbitrarily replaced by two vertical forces acting at the same points and so proportioned that their resultant is the line just spoken of. All external forces are then vertical, there are no inertia forces, and the resultants of the upward and downward forces both pass through the center of gravity and there is complete equilibrium. This is manifestly an easy case to deal with, and the stress diagram in Case II is much simpler than that in Case I, where the external forces can not all be represented on a single straight line. The simplified method has been used by the airplane engineering department of the Air Service.

It is evident that the entire omission of the horizontal components of the chassis strut forces will have a considerable effect on the stresses in the longerons in the bays between the points of attachment of the two struts. This is particularly noticeable in a machine like the JN, where the upper ends of the chassis struts are widely separated and their slope is small. An inspection of the tabulation of the results of the analyses shows that the stresses in the top longeron near the rear of the body are greater for Case II, those in the bottom longeron for Case I. This is natural, as the inertia components of the loads, acting from the free end toward the supporting reactions, tend to increase the compression in the lower longeron and counterbalance the tension in the upper one. The stresses in the struts and wires are nearly the same for the two cases except in the bays between the points of attachment of the chassis struts. This, again, is what might be expected, as, the longerons being nearly parallel and horizontal, the primary duty of the wires is to carry the shear due to vertical loads, while the strut compressions are almost exactly equal to the vertical components of the stresses in the adjacent wires. Strut and wire stresses are therefore substantially unaffected by horizontal components of load at the panel points. In no case, except in the bays between the chassis struts and in a few other bays of the top longeron, is the difference of stress in a member for the two cases as much as 5 per cent. The percentage difference is large in some of these cases, but only where the magnitude of the stresses is small and where the factor of safety would be sure to be very large. The simplified method of Case II leads to an overestimation of the top longeron stresses, as compared with Case I, by about 100 pounds in one bay. The important differences come in the bays between chassis struts. The type of loading used in Case II is manifestly wrong for these bays, and the inclusion of the horizontal components of the strut reactions changes the magnitude of the stresses very radically. In the case of the lower longeron, the effect of these components is to change the stress from a large compression to a tension. The difference between the two cases in the upper longeron is much smaller but the simplified method is not on the safe side. In the struts, the stress given by the simplified method is too high in the member directly over the rear chassis strut, and too low in all others. The only pair of wires much affected is that in the rear bay, where different wires are in tension in the two cases.

Summing up, it is evident that the method of Case II is satisfactory for the rear of the body, but that it gives results very badly in error for some of the members in the neighborhood of the chassis attachment. The simplified loading can well be used for a preliminary analysis to assist in estimating the sizes of members, but it should not be considered as satisfactorily covering the landing loads by itself. When it is employed there should be added to the stresses in the bays of the lower longeron between the chassis struts an amount equal to the horizontal component which would exist in the adjacent chassis strut if the loading of Case I were used. This horizontal component can be found by drawing a triangle of forces. (The addition is algebraic, compressions being taken as negative, and the effect usually is to change the sign of the stresses.)

Case III (ground friction balanced by thrust) gives results identical with those of Case II for all members behind the rear chassis strut. The effect of thrust can be satisfactorily allowed for by adding (algebraically) one-sixth of the weight of the machine to the stress in every bay of each longeron from the nose to the front chassis strut, and one-tenth the weight to the stresses in the longerons between the chassis struts. If the engine is carried in such a way that the thrust can only be transmitted to the fuselage at the rear of the bearers the addition to allow for thrust load can, of course, be omitted forward of this point.

Case IV relates to a three-point landing. The largest dynamic loads usually occur in this type of landing, as a pilot having a forced landing on rough ground will land at as low a speed as possible, and the tail-skid will therefore come into contact with the ground before the shock absorbers on the wheels have had time to extend fully. The only case in which a large dynamic load is likely to occur on a high-speed landing (I and II) is that in which the pilot is unskilled or landing in the dark and fails to flatten out early enough. In considering the landing with tail low all the loads have been taken perpendicular to the top longeron. The effect of friction and of inertia loads is thus allowed for, as the perpendicular to the top longeron makes an angle of about 10° with the true vertical when the machine is resting on the ground. To be strictly accurate, the reactions should be drawn with different inclinations, as the coefficient of friction of the tail-skid is much greater than that of the wheels, but the effect of allowing for this difference would be too small to warrant the additional complication. Since the point of contact of the tail-skid is some distance behind its point of attachment, the reaction of the tail-skid has a moment about its hinge. This is allowed for in the way in which it is actually taken through the hinge and shock-absorber cord in the machine, by putting on equal and opposite horizontal forces at the top and bottom of the strut where the tail-skid is attached. Since the pull in the shock-absorber cord has a vertical component, there should be added another pair of vertical forces, equal and opposite. These have been omitted from the diagram, as their only effect is to increase the compression in the strut over the tail-skid.

The tail-skid load reverses the stresses in the longerons behind the rear chassis strut and changes the diagonal which takes tension in several cases, as compared with the loading arrangements which have already been considered. The stress in the front chassis strut and in the fuselage strut directly opposed to it are also somewhat greater in a three-point than in a two-point landing, because of the backward inclination of the reaction causing the front strut to carry a smaller share of the total load in the latter case than in the former.

The conclusion is that landing stresses should be analyzed for three-point landings (Case IV) and for two-point landings with inclined ground reaction and inertia loads (Case I) and that allowance should be made for the effect of thrust on the stresses in the forward bays of the longerons, especially the upper one. If only one stress diagram is to be drawn for landing loads it should be for the case with tail-skid reaction, as the analysis for flying conditions is very similar in most particulars to that for a two-point landing, and gives stresses of the same order in the rear of the body, whereas there is no other type of loading which resembles that encountered when landing with tail down closely enough to be substituted for it.

FLYING LOADS.

It is customary, in analyzing the fuselage stresses under flying conditions, to consider only the rear portion of the structure, and to treat this as a cantilever loaded in accordance with some set of assumptions and supported either at the rear wing spars or the lower spars. The first assumption as to the support is the more common, but the United States Navy specifications call for an analysis with the reactions taken at the lower wing hinges, which appears to be more logical.

It is very desirable that the analysis under flying conditions be extended to cover the whole structure, from nose to tail. While this is not possible by an extension of the ordinary method with an arbitrarily fixed tail load, the stress diagram failing to close if it is carried on past the wing hinges to the nose of the airplane, a closed diagram can always be secured if full account is taken of all the forces acting on the airplane and of its motions under those forces.

The downward loads on an airplane fuselage in flight may be divided into dynamic loads and tail loads. The dynamic loads are due to the acceleration of the center of gravity of the machine (usually an acceleration parallel to the normal axis of the machine, arising in flattening out after a dive or in some similar sudden change of course). Since the lines of action of the dynamic loads on all parts of the machine at a given instant are parallel so long as there is no angular acceleration of the airplane about its C. G., and since the dynamic loads at the panel points are proportional to the weights concentrated at those panel points the resultant of these loads always passes through the C. G. of the machine.

The tail load, in turn, may be divided into three parts. First is the load, usually downward, required to give static equilibrium about the C. G. This part of the load usually has, for any particular machine, a maximum value of about 4 per cent of the wing loading, its exact magnitude depending on the position of the center of pressure of the wings with relation to the center of gravity. For tail surfaces of normal size, this gives a unit loading of about one-quarter of the unit wing loading. The second part of the tail load goes to overcome the damping moment (M_q) due to the body, wings, and chassis. This moment is about 18 per cent of the total damping moment due to the action of the air against the pitching airplane.[1] The manner of its distribution is uncertain, and in these illustrative examples it was arbitrarily distributed among the various panel points in such a way that the applied forces would sum up to zero, that the sum of their moments about the C. G. would be equal to 18 per cent of the computed total damping moment, and that the loads would be approximately proportional to the distances of their points of application from the C. G. The tail load due to damping appears only when the airplane is rotating about its C. G. (with respect to axes fixed relative to the earth). Finally, there may be an excess of tail load above the sum of these two components, and this excess acts to impart an angular acceleration to the airplane.

In considering tail load, the component of damping due to the tail (82 per cent of the total) must be subtracted from the apparent load to get the true value of the total force acting. The total tail load in any machine at a given instant is then a function of elevator setting, angle of attack, speed, and angular velocity.

It was just demonstrated that two of the three portions of the tail load can be so balanced as to give static equilibrium, the first by the wing load acting at a certain distance from the center of gravity, the second by properly distributed damping loads at the panel points. To balance the third component, the excess producing angular acceleration, inertia loads must be applied much as in the case of the deceleration when landing, the difference being that in this case the structure is treated for the moment as though it had no motions except one of rotation about the C. G., and the inertia load at each panel point therefore acts perpendicular to the line connecting that panel point with the C. G. The ratio of the inertia load to the static load at any panel point is directly proportional to the distance from the C. G. Since the dynamic loads and each component of the tail load taken separately can be balanced by properly allowing for the accelerations, it is evident that any combination of loads can be so balanced, and that it is possible to draw, for any set of fuselage loads which can occur, a closed stress diagram taking in every member of the fuselage and with the wing reactions applied as they actually are in practice.

The dynamic loads can only attain their maximum value when the angle of attack is large, and it therefore requires a perceptible interval, after the elevator is pulled up to flatten out from a dive, for the lift on the wings to reach its highest point. The total down load on the tail, on the other hand, reaches its maximum instantly, as the down load is largest when the

[1] Third Annual Report of the National Advisory Committee for Aeronautics, p. 335: Washington, 1918.

angle of attack is small, when the speed is high, and when the angular velocity and damping moment are small, and all of these conditions are best fulfilled when the airplane is diving and the elevator has just been pulled up. It is then evident that the tail loads and dynamic loads can not reach their maxima at the same time, and any analysis based on the assumption of large loads of both types simultaneously applied is unduly severe.

Furthermore, any analysis in which a maximum down load is applied on the tail and balanced only by reactions at the wing hinges is unduly severe. When there is a large tail load there always exists an angular acceleration, and this produces inertia loads which act upwardly on the rear part of the body, directly balancing the tail load. It can not be too strongly emphasized that the structure as a whole will always be in statical equilibrium if the inertia and damping loads are properly applied, and that there is something wrong with any set of loading assumptions which permits of the existence of an unbalanced moment about the center of gravity.

In connection with the present report the typical fuselage has been analyzed for three different types of flying loads. Cases V and VI were carried through in accordance with the methods just discussed, all inertia loads being included. Case V relates to the maximum dynamic load, Case VI to the maximum tail load. In Case VII an analysis was made for the rear position of the body in accordance with the common assumptions of a dynamic factor of 5, a tail load of 25 pounds per square foot, and the rear of the body acting as a cantilever supported at the rear wing spars.

The fundamental data for Cases V and VI were based on an analysis of the behavior of a JN in a dive and loop, an analysis the most important results of which were published in the Bulletin of the Airplane Engineering Department, United States Army. In connection with the discussion there printed the effect of inertia loads arising from angular acceleration was taken up, but no attempt was made to carry through a complete analysis of the stresses under flying conditions. The assumptions on which that analysis was based were rather too severe, as the maximum dynamic factor by computation was 8.14, whereas it has been shown by accelerometer tests that the dynamic factor seldom rises above 4 and apparently never reaches 4.5. A dynamic factor of 5 was taken as the standard in these illustrative problems, just as for the landing loads, and the computed lifts, tail loads, accelerations, etc., were therefore reduced in the ratio $\frac{5.00}{8.14}$. This procedure is not entirely justifiable, but it gives as good results as can be attained without another complete analysis of a dive.

Case V relates, not to the exact instant of maximum dynamic load, but to the time, 0.42 second after pulling up the elevator, and 0.1 second before the peak of the dynamic load curve is reached, when the angular acceleration is zero. This very much simplifies the analysis by eliminating the "rotative inertia" loads in this case, and has little effect on the total stresses, as the dynamic load at the instant when the angular acceleration disappears is only 4 per cent below its maximum value. The angle of attack of the wings, in this case, was about 12°, and the resultant force on the wings was therefore very nearly perpendicular to the chord. The total lift was divided among the four wing hinges on each side, the lower hinges taking about 85 per cent of the total lift because of the transmission of lift from the upper wing through the inner lift cables to the lower hinges. The distribution between the front and rear hinges depends on the position of the center of pressure. The tail load vector was arbitrarily drawn parallel to that for wing load, although the L/D for the tail is less than that for the wings. The stresses in the fuselage would be very little affected by changing the slope of the tail vector.

The normal tail load for the speed, angle of attack and elevator angle existing at the instant to which Case V relates would be 10 pounds per square foot if there were no angular velocity. The attribution of 82 per cent of the damping moment to the tail and the deduction of the appropriate amount from the tail load reduces this by about 70 per cent, so that the actual tail load corresponding to a dynamic factor of 5 is 2.5 pounds per square foot. This figure would be larger in a machine where the center of gravity was farther forward relative to the wings, but it would never exceed 10 pounds per square foot. No attempt was made to distribute the drag of the body, the whole amount being combined with the wing drag and applied at the hinges. There, again, the procedure is not rigorously correct, as most of the fuselage drag should act at the nose, but the difference between the exact and approximate distribution of fuselage drag would be negligible.

Thrust and torque loads do not appear in any of the analyses under flying loads, as the peed in a limiting dive is so high that there could not be any thrust or very much torque, the propeller tending to operate as a windmill rather than as a propeller. The thrust and torque loads appear in normal flight, and may be important enough to require special analysis in high-powered airplanes with engines having a very low weight per horsepower. There is a gyroscopic torque when flattening out from a dive, and this modifies the stresses in the fuselage members lying in horizontal planes, including the longerons, but it will not be considered in this report.

The analysis in Case VI is very complex, as the loads act in every conceivable direction. They divide, in general, into three groups. First are a set of loads due to translational accelerations. These all act along parallel lines, and, since the lift at the instant after the elevator is pulled up is insignificant, their lines of action are very nearly parallel to the flight path, the only acceleration which amounts to anything being the negative acceleration along the path, due to the excess of total drag over the component of gravity along the path. The second group is made up of the inertia loads due to angular accelerations. Since the lines of action of these loads are perpendicular to the lines connecting their respective panel points with the center of gravity, no two of them act in the same direction, and the drawing of a load diagram becomes a rather complicated and tedious operation. Lastly, there are the external loads, due to air pressure, which balance all these inertia loads. The total load on the tail is roughly 31 pounds per square foot, and its component perpendicular to the flight path is approximately equal to the corresponding component of the force on the wings, so that the resultant is, as already noted, parallel to the flight path. The vector of force on the wings at the very small angle of attack existing during the dive ($-3\frac{1}{2}°$ for this particular machine) is inclined at about 45° to the perpendicular to the chord, and there is therefore a large component of this force which must be taken by the drag bracing. This bracing system has several redundancies, as there are three external drag wires on each side of the fuselage, and there is also a possibility that the drag may be carried through the stagger wires to the lower wing and thence directly to the fuselage at the lower wing hinges. The distribution of the drag between these four alternative routes is necessarily uncertain, depending largely on the initial tension in the several members, but it is absolutely certain that a large share of the drag is transmitted through the external bracing to the nose of the fuselage in a machine which, like the JN, has no stagger wires in the center section. In Case VI it was assumed that 90 per cent of the drag was taken through the external bracing and 10 per cent at the wing hinges, and that the drag taken by the external bracing was divided equally between the upper and lower wires. From the structural standpoint it is desirable to take as much drag as possible through the upper drag wires, as these wires also carry some lift directly to the nose of the fuselage, greatly reducing the stresses in the forward portion and also reducing the stresses in the inner bay of the wing truss.

There is no damping load to be deducted from the tail load in Case VI, as the machine can not acquire an angular velocity instantaneously when the tail is pulled up.

A comparison of the stresses in Cases VI and VII shows that the ordinary method, neglecting all inertia loads, gives stresses far higher than those which can actually prevail, the overestimation in some cases being as much as 200 per cent.

Examination of the tabulated stresses for the first six cases shows that the maximum stress in most instances occurs in Case IV, V, or VI. The maximum comes in Case I in a few instances, but the stress in Case I never exceeds by more than 20 per cent the maximum among IV, V, or VI. As already noted, a dynamic load factor of 5 in a tail-high landing would be a very rare occurrence. Case III also shows a few maxima, but only in the forward three bays, and these bays can be taken care of by adding in a correction for thrust in the manner explained in the first part of this paper. Cases IV, V, and VI, then, may be considered as furnishing a complete and satisfactory analysis. Furthermore, since the maximum stresses occur in Case VI only in the three rear bays of the fuselage it is sufficient, if the work is to be reduced to a minimum, to start the stress diagram for the condition of maximum tail load at the tail post and carry it through the three or four rearmost bays.

Since the complete analysis of a dive has been carried through for only one machine, it is necessary to make certain assumptions as to the magnitude of the damping moments in order to determine the load distribution analogous to Case V for a new machine. The angle of attack for a dynamic load factor of 5 may be taken as 12°, and the tail load required to balance the moment of the wing load about the C. G. can then be computed. Two pounds per square foot, of tail surface should be added to this to allow for damping, and static equilibrium restored by modifying the dynamic loads in the manner already described.

Cases analogous to VI require fewer assumptions, as there is no damping. The total tail load can be determined from a wind tunnel test of a model with the elevators turned up or approximated from the results of such a test on a similar machine. The tail load required to balance the wing moment can be computed as before, the angle of attack being computed for a 45° dive at limiting speed. All the rest of the tail load goes to produce angular acceleration, and is balanced by inertia loads applied in the manner already described. The distribution of the drag will differ in different types of machines, and no definite rules can be laid down, but caution should be exercised not to make the pull in the upper drag wires too large, as the apparent stresses are reduced by so doing. It is better to keep on the safe side and take an excessive proportion of the drag in the lower wires and at the wing hinges.

These notes on loading also furnish some suggestions for the making of sand load tests on fuselages. It is evident that a load of 20 pounds per square foot on the tail, whether or not it is accompanied by a dynamic load, produces larger stresses than would be experienced in practice, and a sand load for maximum tail load would correspond more nearly to actual conditions if upward loads were applied by ropes passing over pulleys and attached at the panel points, to represent the effect of "rotative inertia." If a single sand load test is to be made, however, it would be better to have the loading correspond to the conditions of Case V, with some increase in the tail load to bring the stresses in the rear bays nearer to the maxima found in Case VI. The specifications would then be:

Tail load of 12 pounds per square foot corresponding to a dynamic load of 5, these loads being increased in the same proportion until failure. (These loads are for a training machine. For airplanes having a normal wing loading of over 6 pounds per square foot, the tail load corresponding to a dynamic factor of 5 should be increased, reaching a maximum of 18 pounds per square foot when the normal wing loading is 10 pounds per square foot.)

The loads at panel points not to be exactly equal to the product of the dynamic factor by the weight concentrated at the panel point, but to be increased forward of the center of gravity and reduced behind the C. G. to allow for damping effect on the body and appendages (not including the tail unit). The percentage of increase or reduction should be proportional to the horizontal distance from the C. G. to the point of application of the load, and the total moment about the C. G. of these changes of load should be such that, when the fuselage is supported at the lower wing hinges, the supporting reactions will be in inverse proportion to the distances from the hinges to the center of pressure of the wings at an angle of attack of 12°. This will do away with the difficulty now experienced when a large tail load is applied in a sand load test without any allowance for damping or inertia loads. The fuselage tends to pivot about the rear support, and it is necessary arbitrarily to add a large concentrated load at the extreme nose in order to prevent local failure at the rear support.

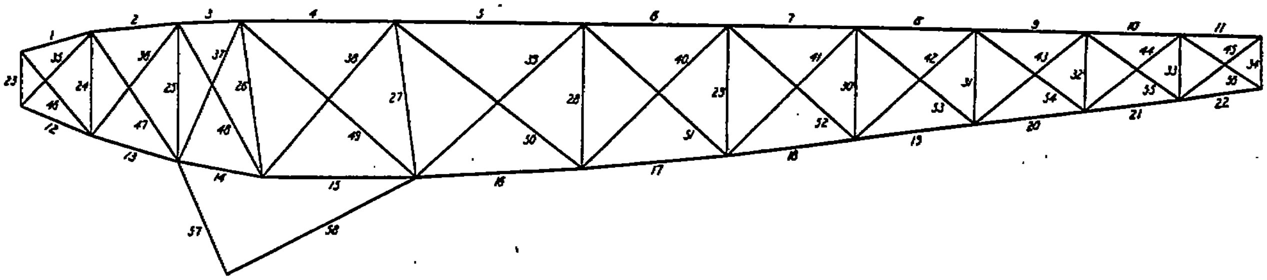
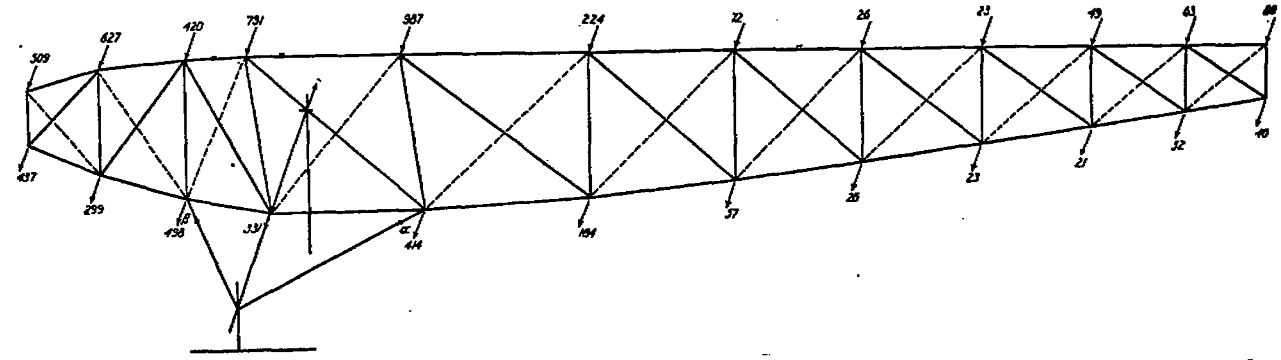
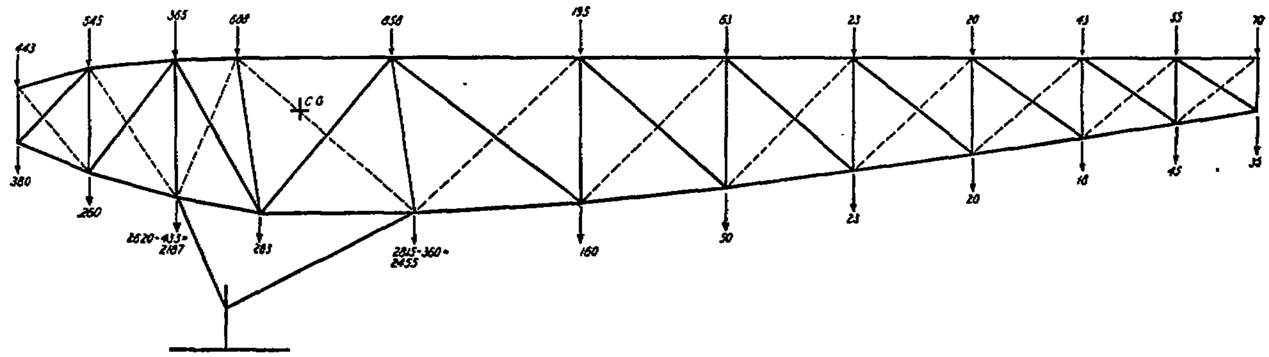


Table of Stresses.

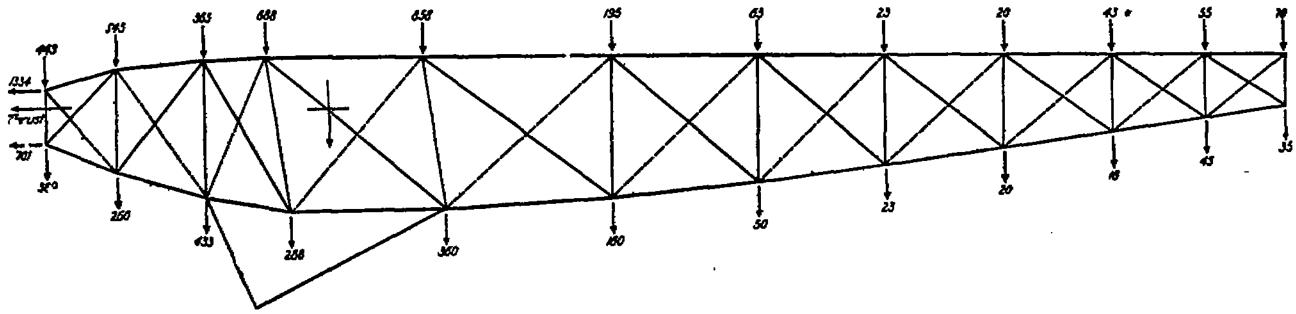
Number.	Case I.	Case II.	Case III.	Case IV.	Case V.	Case VI.	Case VII.
1	+200	0	+1392	0	+17	−92	....
2	+1114	+580	+1840	+580	+590	+44	....
3	+1466	+1195	+2200	+505	+1553	+206	....
4	+1797	+1305	+2123	−712	+2194	+431	....
5	+1420	+1435	+1435	−425	+1519	+757	+4330
6	+1070	+1074	+1074	−523	+1189	+792	+3620
7	+837	+839	+839	−533	+946	+763	+3063
8	+591	+606	+606	−533	+709	+707	+2470
9	+324	+358	+358	−488	+446	+606	+1745
10	+94	+133	+133	−188	+189	+429	+918
11	−29	0	0	0	+5	+89	0
12	−549	−625	+223	−625	−661	−209	....
13	−1540	−1513	−774	−1513	−1531	−313	....
14	+524	−1500	+880	+448	−2375	−400	....
15	+1177	−1980	+1305	+1133	−2841	−680	....
16	−2766	−2343	−2343	−98	−2430	−760	−5793
17	−1688	−1540	−1540	+425	−1553	−760	−4355
18	−1260	−1086	−1086	+528	−1191	−746	−3650
19	−991	−849	−849	+493	−968	−709	−3098
20	−714	−611	−611	+380	−728	−648	−2498
21	−431	−364	−364	+148	−462	−543	−1768
22	−166	−134	−134	−134	−199	−355	−928
23	−457	−450	−55	−450	−447	−470	....
24	−1260	−1088	−1268	−1088	−1093	−121	....
25	−3089	−2043	−2740	−3200	−1564	−182	....
26	−1089	−490	−944	−1430	−2587	−265	....
27	−1786	−2363	−1573	−1278	−1064	−136	....
28	−586	−511	−511	−283	−503	+3	−843
29	−309	−273	−273	−43	−260	−12	−550
30	−237	−206	−206	−23	−212	−38	−495
31	−220	−196	−196	−60	−205	−64	−533
32	−220	−194	−194	−138	−215	−100	−608
33	−160	−141	−141	−333	−177	−180	−6438
34	−77	−70	−70	−70	−122	−294	−695
35	+954	+820	+468	+820	+827	+100	....
36	+1717	+1456	+1555	+1456	+1492	+194	....
37	0	0	0	0	+1188	+252	....
38	0	+1045	0	0	0	+337	....
39	0	0	0	0	0	+36	....
40	0	0	0	0	0	0	....
41	0	0	0	0	0	0	....
42	0	0	0	+58	0	0	....
43	0	0	0	+145	0	0	....
44	0	0	0	+293	0	0	....
45	0	0	0	0	0	0	....
46	0	0	0	0	0	0	....
47	0	0	0	0	0	0	....
48	+529	+585	+1195	+1860	0	0	....
49	+377	+0	+288	+1065	+790	0	....
50	+1283	+1134	+1134	+655	+1109	0	+1825
51	+543	+475	+475	+130	+478	+19	+950
52	+360	+319	+319	+8	+301	+45	+740
53	+337	+300	+300	0	+305	+75	+758
54	+337	+303	+303	0	+324	+127	+893
55	+297	+273	+273	0	+311	+220	+1003
56	+171	+158	+158	+163	+234	+408	+1093
57	−4489	Ind.	−3900	−4545	....	....	....
58	−4629	Ind.	−4105	−2070	....	....	....



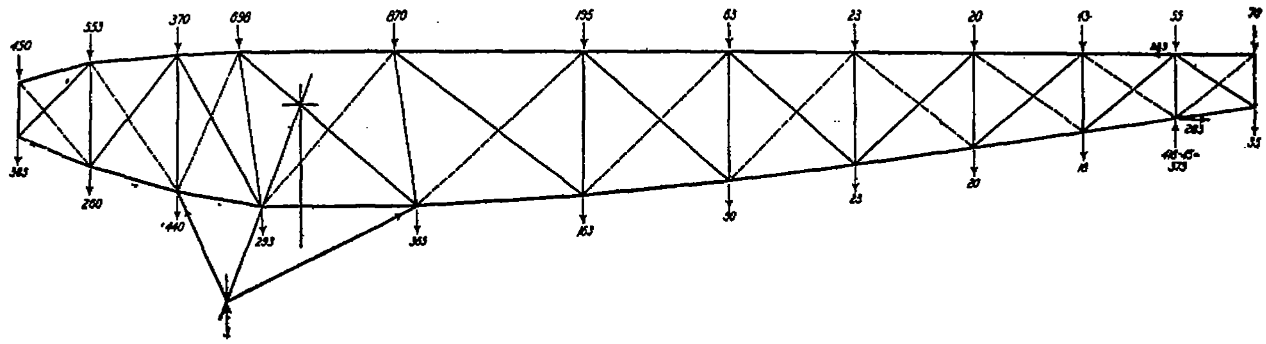
Case I.



Case II.

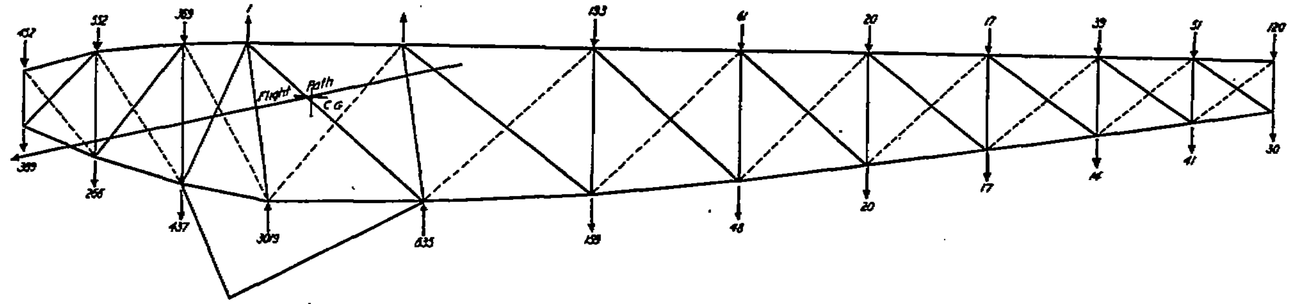


Case III.

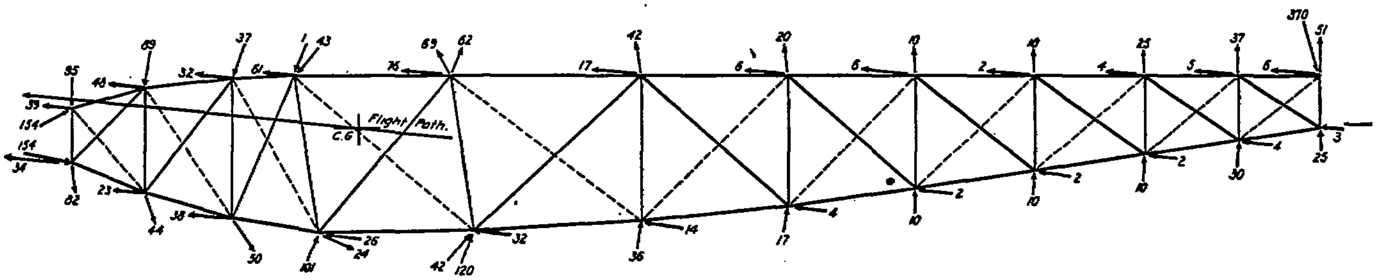


Case IV.

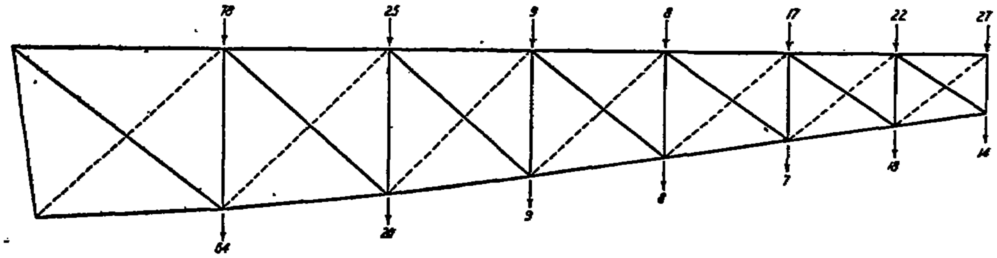
FIGURE 1.



Case V.



Case VI.



Case VII.

FIGURE 2.